Initial Weight Estimation of Twin-Fuselage Configuration in Aircraft Conceptual Design

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Abstract: The Ultra-High Aspect Ratio Wing (UHARW) concept can improve the aircraft's aerodynamic efficiency and reduce fuel consumption. The Twin-Fuselage (TF) configuration is one of the most promising concepts for the UHARW design to reduce the wing bending moments and shear forces. This paper presents the development of a semi-empirical method for the weight estimation of TF aircraft in the initial sizing stage. A physics-based wing weight estimation method is improved for higher aerodynamic analysis fidelity and composite materials, which is used in the design of experiments and the results are applied for regression analysis to establish a semi-empirical method. Eventually, the established semi-empirical weight estimation method is integrated into a TF aircraft conceptual design and performance analysis framework, and a mid-range TF aircraft and a long-range TF aircraft are designed and sized to illustrate its application and efficiency in rapidly estimating the TF aircraft weight breakdown.

Keywords: Twin-fuselage configuration, initial weight estimation, vortex lattice method, composite materials, aircraft conceptual design

1. Introduction

NASA and European Commission have put forward stringent sustainability goals for the next-generation transport aircraft in recent years, including significant reductions in CO2, NOx, noise, etc [1, 2]. In addition, an unexpected revolution in air transportation is needed to recover the aviation industry, which has been affected seriously by the unexpected COVID-19 pandemic, to restore its competitiveness, and to

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address future climate goals. Accordingly, a lot of research is underway, including novel configurations [3], advanced airframe technologies [4], high-efficient and low-cost design methods [5], etc., to achieve a step-change in aircraft performance. One of the most promising solutions to achieve this purpose is the Ultra-High Aspect Ratio Wing (UHARW) configuration, which can improve aerodynamic efficiency and reduce fuel consumption significantly [6].

However, the bending moment and shear force in the UHARW structure is significantly larger than that of the conventional aircraft wings. The UHARW structure needs to be strengthened to carry the larger loads, which results in a significant increase in the wing weight. Therefore, some novel concepts need to be used to lift the wing bending moment and shear force, so as to maximize the benefits of the UHARW concept.

Robust- and sustainable-by-design ultra-higH aspEct ratio wing and Airframe (RHEA) is a European Union-funded project within the Clean Sky 2 Joint Undertaking (rhea-cleansky2.org/). The RHEA research team, composed of TU Braunschweig (DE), University of Strathclyde (UK), Imperial College (UK), DNW Wind Tunnels (NL), and IRT-Saint Exupery (FR), aims to design next-generation passenger aircraft with ultra-high aspect ratio wings and associated airframe to improve aircraft fuel efficiency. To this end, the overarching objective of RHEA is to improve the aerostructural design and efficiency of UHARW by combining advanced numerical and experimental methods for Multidisciplinary Design and Optimization (MDO).

The twin-fuselage (TF) configuration can significantly reduce the wing bending moment and allow for a lighter wing weight by replacing the large mass of the centrally positioned fuselage with two masses positioned outboard [7]. Besides, since the skin thickness of a pressure cabin is proportional to its volume, a reduction in the fuselage weight can be expected due to the smaller individual fuselage diameter of the TF configuration [8]. Therefore, the TF configuration is considered as one of the solutions to utilize the UHARW design in the RHEA project. The TF configuration has been successfully researched and implemented on large transonic airplanes and unmanned aerial vehicles (UAVs), demonstrating the maturity of related technologies. For example, Scaled Composites developed a TF aircraft Stratolaunch carrier aircraft for an air-launch space access system, which is the largest all-composite aircraft ever built

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[9]. Virgin Galactic developed a four-engine TF aircraft WhiteKnightTwn to launch the sub-orbital vehicle SpaceShipTwo [10]. The German Aerospace Center (DLR) has developed and flown a 4-seat TF aircraft HY4, which is powered by a hydrogen fuel cell system [11]. Ma et al. [12] developed and flight-tested a TF aircraft with distributed electric propulsion. Moreover, there are many studies on TF transport aircraft. Lockheed and NASA performed the conceptual design and wind tunnel experiments for a large cargo TF aircraft in the same class as the Boeing 747 transport aircraft and C-5A [13] and conducted the simulator study of its flight characteristics during approach and landing [14]. Chiesa et al. [15] presented a TF aircraft preliminary design study. Vedernikov et al. [16] analyzed the advantages of the TF configuration through a design case based on the prototype of A320.

However, these studies on the TF aircraft design neither introduce the complete design framework and process, nor present the conceptual design methodology, which is challenging to be used as a reference in the TF aircraft design. Besides, weights estimation is particularly important in conceptual design due to its notable impact on the overall aircraft performance. Udin et al. [17] developed a semi-analytical wing mass estimation method for the TF aircraft configuration, which is a class II & 1/2 wing mass estimation method and requires detailed wing design parameters. In contrast, a semi-empirical weight estimation method that requires fewer inputs and has acceptable accuracy is more suitable for the early design stage of TF aircraft. Semi-empirical weight estimation methods are presented in many aircraft design handbooks for conventional aircraft [8, 18]. These methods are either too complex for the initial sizing of the TF aircraft or are applicable only for conventional configurations. Therefore, a semi-empirical weight estimation method for the TF configuration applicable to the initial sizing is required.

To this end, a semi-analytical TF aircraft wing mass estimation method is improved in this work, including improved fidelity of aerodynamic load estimation and extensions to advanced composite materials. Design of Experiment (DoE) and regression methods are used to establish a class II wing mass estimation method for TF aircraft based on the modified physics-based method. Finally, the class II method is integrated into a TF aircraft conceptual design framework and two different classes of TF passenger aircraft are designed to illustrate the design process of the TF aircraft and to demonstrate the application of the developed TF aircraft mass estimation method.

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2. Methodology

The weight estimation methodology presented in this section is a combination of the method for the classical aircraft weight estimation method [18, 19] and the semi-empirical wing mass estimation method developed for the TF configuration.

2.1. Initial Weight Estimation

According to the classical aircraft design method [18], the maximum takeoff mass can be expressed as:

$$m_{\mathrm{TO}} = m_{\mathrm{crew}} + m_{\mathrm{pay}} + m_{\mathrm{f}} + m_{\mathrm{e}} \tag{1}$$

where m_{crew} is the crew mass, m_{pay} is the payload mass, m_{f} is the fuel mass, and m_{e} is the empty mass.

In Eq.(1), m_{crew} and m_{pay} can be obtained from the Top-Level Aircraft Requirements (TLAR), and the fuel mass can be calculated by the Breguet range equation which is with respect to the design range from the TLAR. For conventional aircraft, the empty mass is calculated by the empty mass fraction ($m_{\mathrm{e}}/m_{\mathrm{TO}}$), which can be estimated statistically from historical trends developed based on the database. However, this estimation method is only applicable to conventional aircraft and cannot show convincing accuracy for unconventional aircraft, such as the TF configuration, due to the lack of these novel aircraft configuration's real statistical data. Since the most significant difference between the TF aircraft and conventional aircraft is the wing mass estimation method due to the different load distribution, this paper establishes a semi-empirical wing mass estimation method for the TF aircraft applicable to the initial sizing stage, which is presented in subsequent sections.

Since the mass estimation method for other components of TF aircraft is similar to that of conventional aircraft, including fuselage, tailplanes, engines, etc., the classical mass estimation method FLOPS [19] is used. According to the FLOPS, the fuselage mass can be calculated by:

$$m_{\mathrm{f}} = 1.35 \cdot \left(L_{\mathrm{f}} \cdot D_{\mathrm{f}}\right)^{1.28} \cdot \left(1 + 0.05 \cdot N_{\mathrm{e}}\right) \cdot \left(1 + 0.38 \cdot F_{\mathrm{c}}\right) \cdot N_{\mathrm{f}} \tag{2}$$

where L_{f} is the total fuselage length, D_{f} is the average fuselage diameter, N_{e} is the fuselage mounted

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engines' number, F_{c} is the cargo aircraft floor factor (0.0 is for passenger aircraft and 1.0 is for military cargo aircraft), and N_{f} is the fuselages' number.

And the horizontal tail and vertical tail masses can be estimated as:

$$m_{\mathrm{ht}} = 0.53 \cdot S_{\mathrm{ht}} \cdot DG^{0.2} \cdot \left(\lambda_{\mathrm{ht}} + 0.5\right) \tag{3}$$

$$m_{\mathrm{vt}} = 0.32 \cdot S_{\mathrm{vt}} \cdot DG^{0.3} \cdot \left(\lambda_{\mathrm{vt}} + 0.5\right) \cdot N_{\mathrm{vt}}^{0.7} \tag{4}$$

where S_{ht} and S_{vt} are the areas of the horizontal tail and vertical tail, respectively, DG is the designed gross mass, λ_{ht} and λ_{vt} are the taper ratios of the horizontal tail and vertical tail, and N_{vt} is the number of vertical tails.

More details of the other components' mass estimation method, including engines, landing gears, paint, systems, etc., can be found in Ref. [19].

2.2.Wing Mass Estimation Method

This section presents the development of a semi-empirical wing mass estimation method for the TF configuration based on a modified physics-based method.

2.2.1. Improvement of a Semi-Analytical Wing Mass Estimation Method

The most significant difference between TF aircraft and conventional aircraft in the initial sizing is the wing mass estimation approaches due to their different load distribution [16], as illustrated in Fig. 1, which means that the conventional aircraft's wing mass estimation methods do not apply to the TF configuration.

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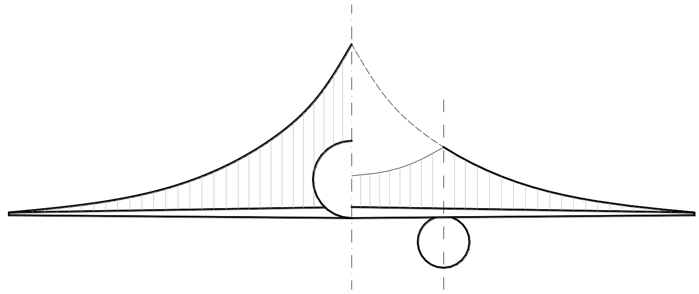

Fig. 1. Wing bending moment comparison between TF configuration (right) and conventional configuration (left) [16].

Udin et al. [17] developed a semi-analytical wing mass estimation method for the TF configuration, estimating the wing structural mass through integrating the wing spanwise mass distribution, including wing structure mass, fuel mass, and concentrated mass such as the engines, with the aerodynamic load. The wing structural relative mass is expressed as

$$m_{\mathrm{s}} = k_{\mathrm{sl}} k_{\mathrm{tw}} k_{\mathrm{man}} \left(m_M + m_Q \right) + m_{\mathrm{rib}} + m_{\mathrm{ail}} + m_{\mathrm{sk}} + m_{\mathrm{flap}} \tag{5}$$

where m_{s} is the relative wing structural mass, k_{sl} is the service life factor, k_{tw} is the twist moment factor, and k_{man} is the manufacturing factor, m_{rib} is the relative rib mass, m_{ail} is the relative ailerons mass, m_{sk} is the relative load free wing skin mass, m_{flap} is the relative flaps mass, and m_M and m_Q are the estimated relative structural mass counteracting the wing bending moment and shear force, which can be calculated as

$$m_M = 2 \frac{\rho n_z g}{\sigma_u T_r} \frac{b^2}{4} E_T \int_0^1 \frac{Mo_{\mathrm{sum}}}{P(y) \cos \Lambda} \mathrm{d}y \tag{6}$$

$$m_Q = \frac{\rho n_z g}{\sigma_{us}} \frac{b}{2} \int_0^1 \frac{Q_{\mathrm{sum}}}{\cos \Lambda} \mathrm{d}y \tag{7}$$

where ρ is the structural material density, n_z is the design load factor, g is the gravitational

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acceleration, σ_u is the ultimate direct stress, σ_{us} is the ultimate shear stress, b is the wing span, Λ is the wing half-chord sweep, Mo_{sum} and Q_{sum} are the total reduced bending moment and the total reduced shear force caused by aerodynamic loads, wing structural mass, fuel mass, and concentrated mass, which can be obtained by integrating the spanwise distributed loads:

$$Mo(y) = \int_y^1 Q(y)\ \mathrm{d}y \tag{8}$$

$$Q(y) = \int_y^1 q(y)\ \mathrm{d}y \tag{9}$$

where $q(y)$ represents the wing spanwise distributed loads, including aerodynamic load, fuel mass, wing structural mass, and concentrated loads.

For the aerodynamic estimation, a linear equation is used in this method to model the wing spanwise distribution of aerodynamic loads, which is given by:

for the inboard wing section ($0 < y < y_{\text{f}}$):

$$q_{\text{a}} = \frac{2}{y_{\text{f}}(1-\lambda)+\lambda+1} \tag{10}$$

and for the outboard wing ($y_{\text{f}} < y < 1$):

$$q_{\text{a}} = 2\frac{\left[(1-y)/(1-y_{\text{f}})\right](1-\lambda)+\lambda}{y_{\text{f}}(1-\lambda)+\lambda+1} \tag{11}$$

where λ is the wing taper ratio, and y_{f} is the fuselage spanwise location.

The wing secondary structures mass can be estimated by existing semi-empirical methods [8, 17]. Some of them are shown below.

The total aileron mass can be estimated by [8]:

$$m_{\text{ail}} = 3m_{\text{ref}}k_{\text{bal}}\left(S_{\text{ail}}/S_{\text{ref}}\right)^{0.044} \tag{12}$$

where m_{ref} is the reference specific weight, which is equal to $56\ \text{N/m}^2$ for Al-alloy skins, S_{ail} is the

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total platform area of all ailerons, S_{ref} is the wing reference area. The factor k_{bal} is 1.0 for unbalanced ailerons, 1.3 for aerodynamic-balanced ailerons, and 1.54 for mass-balanced ailerons.

The mass of leading-edge flaps can be calculated as:

$$m_{\mathrm{LE}} = 4.83 m_{\mathrm{ref}} \left(S_{\mathrm{LE}} / S_{\mathrm{ref}}\right)^{0.183} \tag{13}$$

where S_{LE} is the leading-edge flaps mass.

The mass of trailing edge flaps can be calculated by:

$$m_{\mathrm{ref}} = 1.7 k_{\mathrm{sup}} k_{\mathrm{slot}} m_{\mathrm{ref}} \left[1 + \left(m_{\mathrm{TO}} / m_{\mathrm{r}}\right)^{0.35}\right] \tag{14}$$

where $m_{\mathrm{r}} = 10^6\ \mathrm{N}$, and the factor k_{sup} and k_{slot} represent the flap motion support and the number of slots, which can be found in Ref. [8].

Besides, miscellaneous items' mass should also be considered, which represents the scattered mass components. Since TF aircraft's landing gears could be stowed underneath the fuselage cabin floor, i.e., no extra fairings are required for landing gears, the mass of the miscellaneous items for TF aircraft should be relatively small, weighing less than 1% of the wing mass [8].

The linear approximation method for lift distribution in the wing box structure is not accurate enough for a physics-based wing mass estimation method. Besides, since this method was published decades ago, it does not apply to advanced composite materials. These shortcomings will be improved in the subsequent sections.

a. Aerodynamic Analysis Fidelity Improvement

Aerodynamic loads account for a remarkable proportion of the total wing loads. Therefore, it is necessary to estimate it accurately, otherwise, the estimated wing mass will have a significant error.

In this work, a VLM tool AVL [20] is integrated into this method to estimate the wing spanwise aerodynamic loads (q_{a}) distribution according to the input wing geometries. By integrating q_{a} along the wing span, one obtains the reduced shear force Q_{a} due to the aerodynamic loads. Then integrating Q_{a} along the wing span, one obtains the reduced bending moment M_{a} due to the aerodynamic loads.

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A comparison of the spanwise distribution of q_{a}, Q_{a}, and M_{a} of the VLM and the original linear approximation method is shown in Fig. 2. It can be seen that there is a significant difference in the spanwise aerodynamic loads' distribution between the VLM and the linear approximation method, resulting in different distributions of Q_{a} and M_{a}.

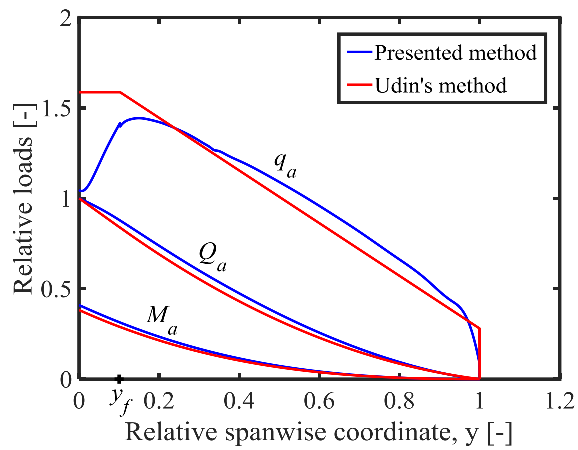

Fig. 2. Spanwise distributions of reduced quantities caused by aerodynamic loads.

b. Improvement for Composite Wings

It is clear that composite materials will be the major material for the next-generation transport aircraft. Therefore, the presented wing mass estimation method needs to be improved. In preliminary design, the cut-off strain method can be used to size the composite wing structures, which are determined according to the worst of all situations [21]. The cut-off strains combined with the 10% rule [22] are used to calculate the allowable for composite materials. Elham et al. [21] presented the laminate properties of composite materials based on the proposed approaches, and the maximum allowable stress in the wing box lower panel can be expressed as

$$\sigma_{\mathrm{max,l}} = E_{\mathrm{x}}\varepsilon_{\mathrm{t}} \tag{15}$$

where E_{x} is the laminate stiffness in the x-direction and ε_{t} is the cut-off tensile strain. The maximum

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allowable stress in the wing box upper panel is given by

$$\sigma_{\mathrm{max,u}} = E_{\mathrm{x}}\varepsilon_{\mathrm{c}} \tag{16}$$

where ε_{c} is the cut-off compressive strain. Besides, the buckling stress can be calculated by

$$\sigma_{\mathrm{max,b}} = 0.725F\left(E_{\mathrm{x0}}^{2}Z\right)^{1/4}\sqrt{\frac{P}{L}} \tag{17}$$

And the maximum allowable shear stress can be expressed as

$$\tau_{\mathrm{max}} = G_{\mathrm{xy}}\varepsilon_{\mathrm{s}} \tag{18}$$

where G_{xy} is the laminate shear stiffness and ε_{s} is the cut-off shear strain.

The required laminate properties in the above equations can be calculated by the 10% rule [22], as follows:

$$E_{\mathrm{x}} = \left(0.1 + 0.9\frac{m}{100}\right)E_{\mathrm{x0}} \tag{19}$$

$$E_{\mathrm{y}} = \left(0.1 + 0.9\frac{100-m-n}{100}100\right)E_{\mathrm{x0}} \tag{20}$$

$$G_{\mathrm{xy}} = \left(0.028 + 0.234\frac{n}{100}\right)E_{\mathrm{x0}} \tag{21}$$

$$\nu_{\mathrm{xy}} = \nu_{\mathrm{yx}} = \frac{1}{1+4\left(\frac{100-m-n}{n}\right)} \tag{22}$$

where m is the plies' percentage at 0 degrees, n is the plies' percentage at ± 45 degrees, and E_{x0} is the stiffness of the unidirectional lamina along the principal direction.

c. Iteration Loops

In this semi-analytical wing mass estimation method, a guess value of the wing structural mass is required as one of the initial inputs. For conventional aircraft, there is a well-known 12% rule [23], i.e., the wing

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mass can be estimated as 12% of the MTOW. The initial input for the wing structural mass can be taken as 12% of the MTOW, and an additional iteration loop is required to compare the calculated wing structural mass with the initial input value. If their difference is less than the convergence criteria, the calculation result of the wing mass is output, otherwise, the wing mass obtained from this iteration is calculated again as the input value until convergence.

d. Validation

The modified semi-analytical wing mass estimation method that improves the accuracy of aerodynamic analysis and extends the composite material model is validated in this section.

The wing mass estimation method presented in this paper is intended for the advanced TF concept. However, so far there are no data available on "real" wing mass for metallic or composite TF aircraft. Nevertheless, the presented physics-based wing mass estimation method is also applicable to conventional aircraft if y_f equals the fuselage radius [17]. In this paper, several metallic medium-range and long-range aircraft and composite medium-range and long-range aircraft were selected for validation, including B737-200 [24], A320-200 [25], B777-200 [26], A330 [27], D8 [1], PFC [28], and B787 [29]. The results are shown in Fig. 3, including results obtained with the original method proposed by Udin et al. [17], the modified method presented in this paper, and the wing mass data from the above-cited references. It can be seen that the accuracy of the modified method fluctuates for the analysis of different aircraft's wing mass, but in general, is more accurate than the original method.

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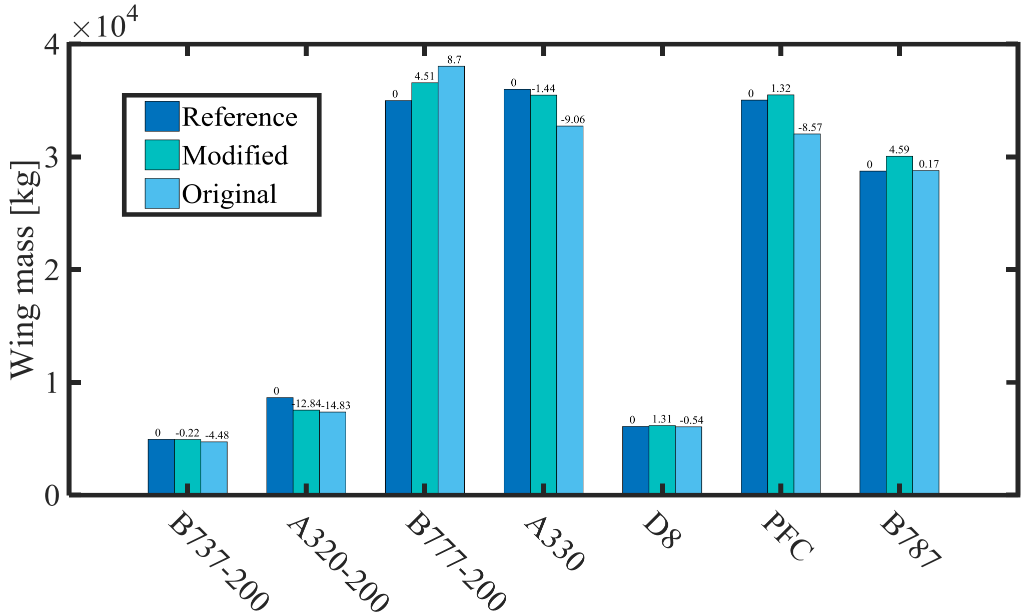

Fig. 3. Validation of medium-range and long-range aircraft with conventional aluminum and CFRP materials ("Original" means the method from Ref. [17]; "Modified" means the presented method in this paper).

2.2.2. Development of a Semi-Empirical Wing Mass Estimation Method

As described, the development of semi-empirical (class II) wing mass estimation methods with sufficient sensitivity to important design parameters is necessary for the conceptual design or initial sizing of unconventional aircraft. In this paper, the semi-empirical methods are developed for four kinds of TF aircraft: MR with metallic materials, LR with metallic materials, MR with composite materials, and LR with composite materials. For each interested configuration, typical design parameters are selected and DoE is performed with the improved physics-based wing mass estimation method. Finally, the estimated wing mass results are used in a multiple linear regression to obtain the semi-empirical equations.

a. Equation Form

The semi-empirical method for the TF aircraft's wing box mass has the following form:

$$m_{\text{wingbox}} = C \cdot m_{\text{TO}}^{E_{\text{m}}} \cdot \left(W/S\right)^{E_{\text{ws}}} \cdot AR^{E_{\text{AR}}} \cdot \left(\cos\Lambda\right)^{E_{\Lambda}} \cdot \left(t/c\right)^{E_{\text{t}}} \cdot V_{\text{m}}^{E_{\text{V}}} \cdot \left(1+\lambda\right)^{E_{\lambda}} \cdot n_{Z}^{E_{\text{nz}}} \cdot \left(1-Z_{\text{f}}\right)^{E_{\text{zf}}} \cdot Z_{\text{e}}^{E_{\text{ze}}} \qquad (23)$$

where the different exponents E and the constant C are used for each configuration, m_{TO} is the

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maximum take-off weight, W/S is the wing loading, AR is the wing aspect ratio, Λ is the wing quarter-chord sweep angle, t/c is the average wing airfoil relative thickness to chord ratio, V_{m} is the maximum operating velocity (m/s), λ is the wing taper ratio, n_{Z} is the maximum positive load factor, Z_{f} is the relative spanwise fuselage location, and Z_e is the relative spanwise engine location.

In addition, the wing secondary structures mass can be estimated using the semi-empirical methods presented in Sec. 2.2.1.

b. Assumptions

Conventional aluminum structures made from 7075 aluminum are used for the metallic TF aircraft wing box, including covers, spars, webs, and ribs. While the CFRP material is used for the composite TF aircraft wing box, which is built with the graphite-epoxy composite. The wing covers are made of 50% fibers in the center axis direction, 38% fibers in $\pm 45^{\circ}$, and 12% fibers perpendicular to the axis [30]. According to the fiber layup approach, the maximum allowable stresses and shear forces in the wing box panels and webs can be estimated using Eqs. (15-18).

Some typical design parameters are included in Eq. (23), but there are many more parameters influencing the wing mass, including wing platform geometry, structure layout, etc. In order to make the established equations applicable to the early design stage, some assumptions are necessary.

The geometry definition of a TF aircraft wing is shown in Fig. 4. It is assumed that the inboard wing section is not swept and tapered and that the inboard/outboard wing sections are joined at the fuselage centerline. The outboard wing section is tapered, and the engine is assumed to be installed on the outboard wing section.

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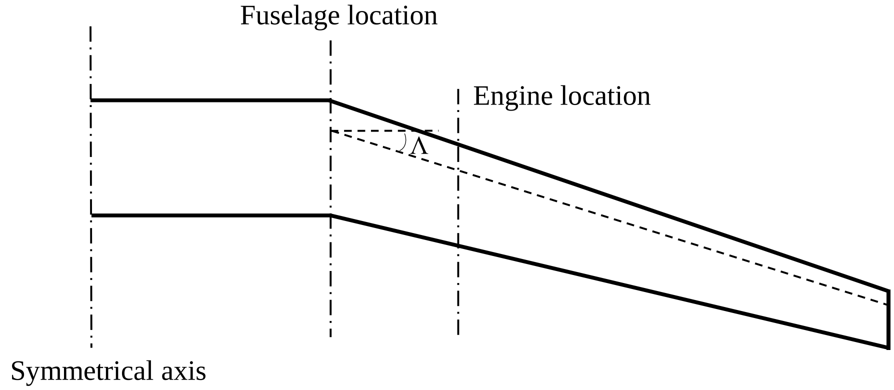

Fig. 4. Wing platform geometry definitions.

The maximum operating Mach number is assumed a function of the outboard wing section sweep angle and wing thickness-to-chord ratio according to the Korn equation [31], as

$$M_{\mathrm{m}} = \frac{k_{\mathrm{a}}}{\cos\Lambda} - \frac{t/c}{\cos^2\Lambda} - \frac{C_{\mathrm{L,des}}}{10\cos^3\Lambda} \tag{24}$$

where k_{a} is the airfoil technology factor, which is taken as 0.93 in this paper and $C_{\mathrm{L,des}}$ is the designed lift coefficient, which is assumed as 0.45 in this paper.

In addition, the ailerons area factor is assumed as 0.06, the flaps area factor is assumed as 0.18, the leading edge flap motion support is assumed to be Fowler flaps with hooked tracks, and the trailing edge flaps are assumed to be double-slotted flaps with articulating vanes.

Most of the presented assumptions are typical for transport passenger aircraft with some minor modifications considering the characteristics of the TF configuration. Some values may be very rough estimates, but this formula is established for the initial sizing purpose providing approximation values for the next stage of high-fidelity wing mass estimation, so these rough estimates are considered acceptable here.

Design of Experiments

DoE is used for regressions to calculate the exponents E and the constant C in Eq. (23). The DoE method of Latin Hypercube Sampling (LHS) is selected to sample the design space since it covers the design space uniformly. One hundred of samples were generated for each configuration, and the lower and

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upper bounds of the design parameters for the mid-range aircraft and long-range aircraft are listed in Table 1, separately.

Table 1. Design variables boundaries

Design variables	Mid-range aircraft		Long-range aircraft	
m_{TO}, kg	40,000	80,000	200,000	300,000
W/S, $\left(\mathrm{N/m^2}\right)$	4,500	6,500	6,500	8,000
AR	15.0	30.0	15.0	30.0
Λ, deg	10.0	25.0	15.0	30.0
t/c	0.1	0.16	0.1	0.16
V_{m}, $(\mathrm{m/s})$	200	260	200	260
λ	0.2	0.4	0.2	0.4
n_{Z}	1.5	2.5	1.5	2.5
Z_{f}	0.1	0.2	0.1	0.2
Z_{e}	0.3	0.4	0.3	0.4

c. Regression Analysis

The 100 DoE samples are calculated for each configuration and material, including the metallic MR-TF aircraft, the metallic LR-TF aircraft, the CFRP MR-TF aircraft, and the CFRP LR-TF aircraft, with the improved physics-based wing mass estimation method. The multiple linear regression was used to obtain the exponents and constants in Eq. (23), and the results are given in Table 2.

In the application of this method, the corresponding data from Table 2 can be selected and substituted into Eq. (23) for the analysis according to the mission type and the structural material.

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Table 2. Wing box constants and exponents for each concept and material

Aircraft		C	E_{m}	E_{ws}	E_{AR}	E_{Λ}	E_{t}	E_{V}	E_{λ}	E_{nz}	E_{zf}	E_{ze}
Al	MR	-12.8809	1.4487	-0.3817	1.3147	-1.0650	-0.8390	0.1452	2.4358	0.9128	2.6202	-0.1853
	LR	-11.5310	1.3841	-0.3424	1.1530	-0.6774	-0.7501	0.1554	2.3890	0.8221	2.5165	-0.1891
CFRP	MR	-13.1132	1.4732	-0.4303	1.3034	-1.0493	-0.8155	0.1359	2.2995	0.9464	2.5861	-0.1729
	LR	-11.9976	1.4127	-0.3628	1.1639	-0.6991	-0.7396	0.1482	2.3003	0.8671	2.5298	-0.1800

d. Validation

The semi-empirical wing mass estimation method developed here is intended for advanced TF aircraft. There is no real data on the TF aircraft. However, the presented wing mass estimation method is also applicable to conventional aircraft and it is possible to validate the wing mass estimation method with data from conventional aircraft. The selected validation data consists of 11 transport aircraft with aluminum or composite constructions.

The validation results are shown in Fig. 5. Elham et al. [21], Lissys Ltd [32], and Peter et al. [28] presented the wing mass data for the A300, B787, and PFC, respectively. While wing mass data for other commercial transport aircraft were extracted from Ref. [33]. It can be seen that the estimation differences (absolute errors) are below 10% for most aircraft and below 15% for all aircraft, which indicates that the accuracy of the presented semi-empirical method developed for the initial sizing of TF aircraft is acceptable.

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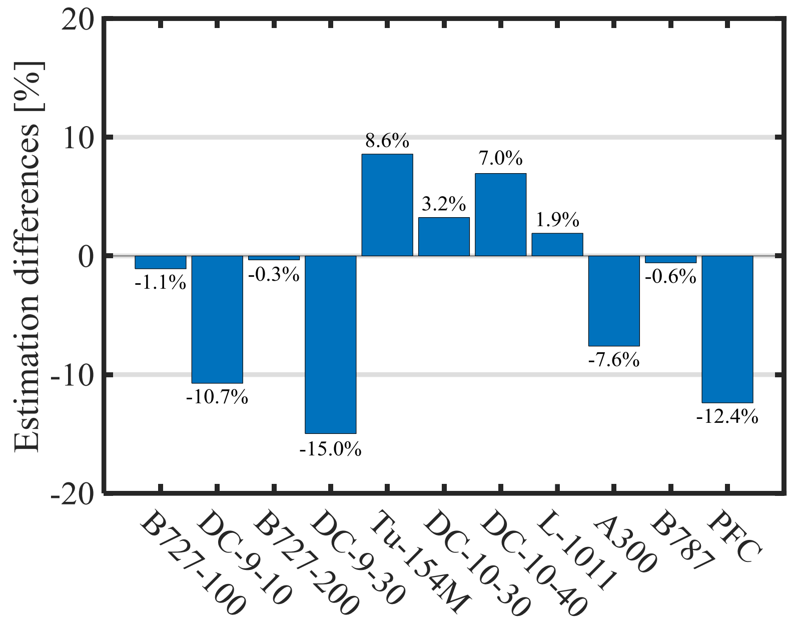

Fig. 5. Validation of the semi-empirical wing mass estimation method for TF aircraft.

e. Impact of Design Parameters

Since the design parameters included in Eq. (23) were selected based on experience and references, sensitivity analysis is used to analyze the importance of each design parameter on the wing mass in order to understand the characteristics of the TF aircraft wing.

The sensitivity analysis in this paper is carried out using a standardized dimensionless form of sensitivity [12]:

$$S_{\mathrm{i}} = \frac{\left(\partial y\left(X\right) / y\left(X\right)\right)}{\left(\partial x_{\mathrm{i}} / \partial x_{\mathrm{i}}\right)} \qquad \mathrm{i}= 1, 2, ..., l \tag{25}$$

A320 and B777 were used as design cases for sensitivity analysis for mid-range and long-range missions, separately. The results are shown in Figs. 6 and 7. Since the same method is used, the importance distribution of each design parameter for MR and LR is basically the same. Maximum takeoff mass and aspect ratio are the most important design parameters because they are the overall parameters that have a significant impact on the wing area. Next are wing thickness-to-chord ratio and maximum positive load factor, which have a strong influence on the wing structural mass, while wing loading, taper ratio, and

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spanwise fuselage location have similar importance for wing box mass.

The sensitivity analysis used here helps researchers identify the most important parameters by comparing the effects of each design parameter. For example, when sizing a TF aircraft wing, one should realize that the maximum takeoff mass and aspect ratio are the most important parameters and that more attention should be paid to them in order to reduce wing weight.

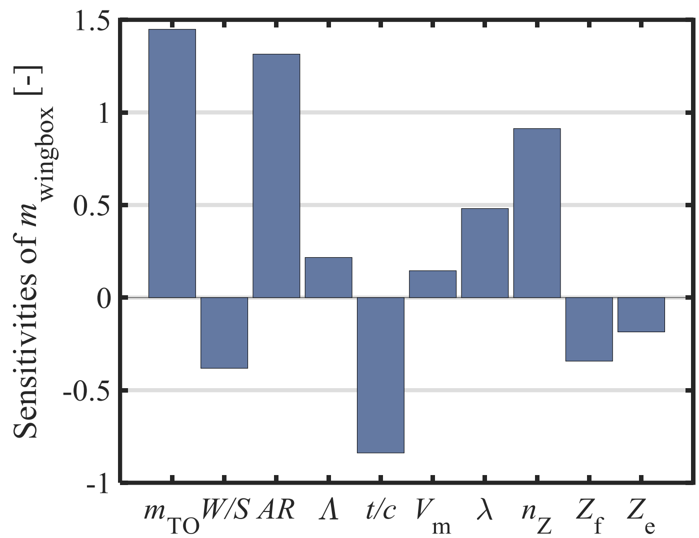

Fig. 6. Sensitivity analysis of wing box mass for mid-range aircraft (A320).

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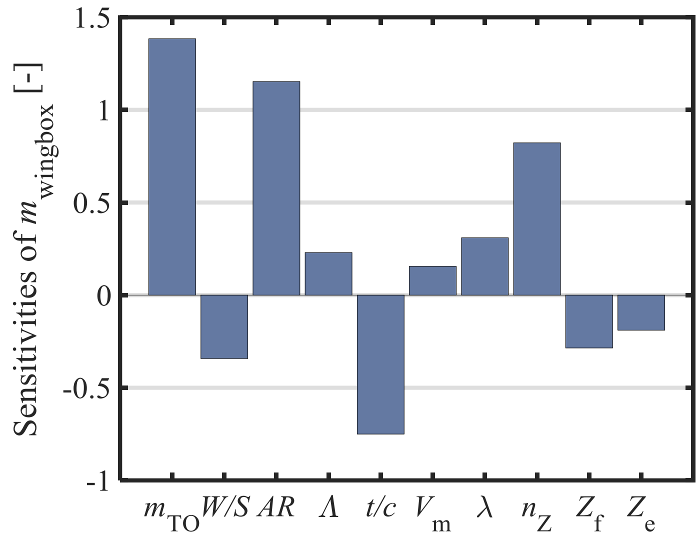

Fig. 7. Sensitivity analysis of wing box mass for long-range aircraft (B777).

2.3. Conceptual Design Framework

A conceptual design and analysis process for TF aircraft has been developed by the authors in Ref. [34], consisting of two main modules: initial sizing and performance analysis. The aircraft initial sizing is performed by using PyInit [6], an in-house tool developed by the authors, to generate the constraints diagrams, size the components, etc. Then the initial sized aircraft is imported into the SUAVE [5], an open-source aircraft performance analysis tool developed by Stanford University, for the analysis of weight breakdown, mission segments, flight performance, etc., through convergence iterations. The semi-empirical wing mass estimation method developed in this paper is integrated into the TF aircraft conceptual design framework, as shown in Fig. 8. The wing mass estimation method in SUAVE is replaced by the presented semi-empirical method for iterative calculations.

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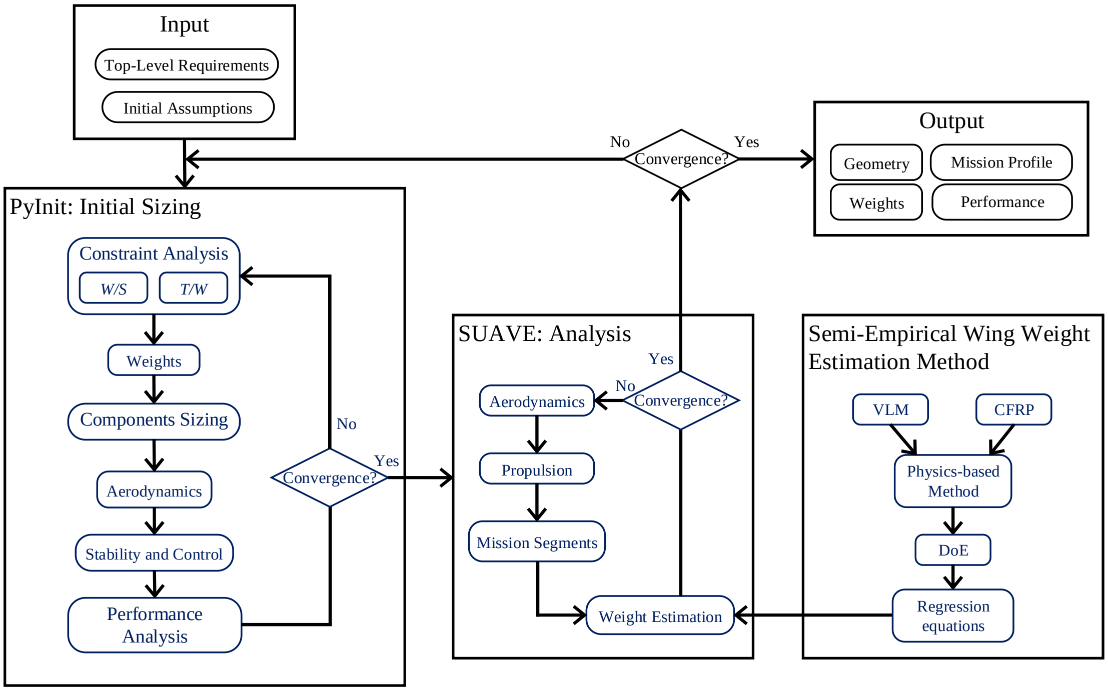

Fig. 8. TF aircraft conceptual design flowchart integrating the developed wing mass estimation method.

3. Design Study

Two design cases, including a Mid-Range (MR) TF aircraft and a Long-Range (LR) TF aircraft, are designed in this section to demonstrate the developed wing mass estimation method and the modified TF aircraft conceptual design framework.

3.1. Top-Level Aircraft Requirements and Assumptions

An MR and an LR twin-fuselage transport aircraft comparable to the A320 and B777 are considered here, separately. The entry-into-service time of these two aircraft is assumed to be 2040, and therefore some novel technology assumptions are introduced, including an assumed 55% laminar flow region of the wing and tails due to hybrid laminar flow control technologies, an assumed 20% reduction in structural mass due to advanced composite materials and structures, and an assumed +1.5g maximum positive load factor due to load alleviation technologies. It should be noted that according to the ICAO airport operation requirements, the wing span and the main landing gear span cannot exceed 36 m and 9 m for the MR-TF aircraft, and 65 m and 14 m for the LR-TF aircraft. The top-level aircraft requirements are given in Table 3.

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Table 3. Top-level aircraft requirements

Parameter	Unit	MR-TF	LR-TF
Reference	–	A320	B777
Cruise Mach number	–	0.78	0.84
Cruise altitude	ft	33,000	35,000
Range	nm	3400	7500
Passengers (2 class)	–	150	350
Approach speed	kt	136	140
Diversion range	nm	200	200
Diversion hold	min	10	10
Contingency fuel	%	3	3
Gate-box limit	m	36	65
Main landing gear span	m	9	14

3.2.Sizing and Performance Analysis

Corresponding to the presented technology assumptions and the top-level aircraft requirements, the constraints diagrams of the MR-TF aircraft and the LR-TF aircraft were sized by PyInit. The constraints diagrams and the selected design points are shown in Fig. 9 and 10.

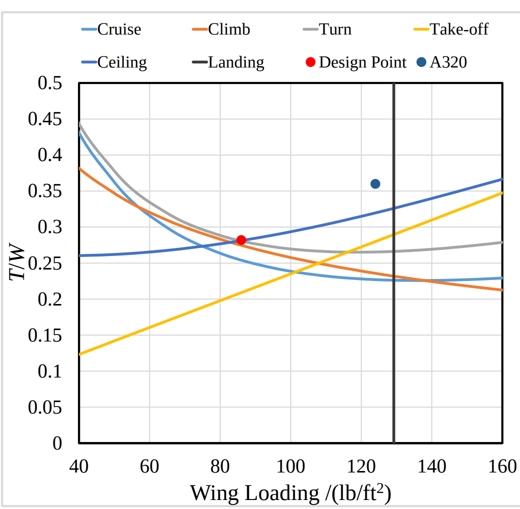

Fig. 9. Constraints diagrams of the MR-

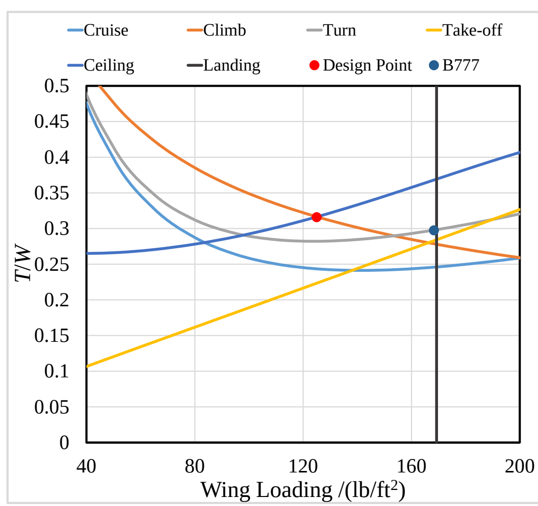

Fig. 10. Constraints diagrams of the LR-

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TF aircraft. TF aircraft.

Since the TF configuration has the advantage of significantly reducing the bending moment in the wing structure, these two TF aircraft were designed with ultra-high aspect ratio wings to improve aerodynamic efficiency and reduce fuel consumption. Therefore, the high wing configuration was selected to meet the required engine and wingtip clearance. Considering the aeroelastic characteristics and avoiding the downwash of the wing and engine outflow, the high slab tail configuration was adopted. The geometric dimensions of the initial sized aircraft are shown in Fig. 11 and 12.

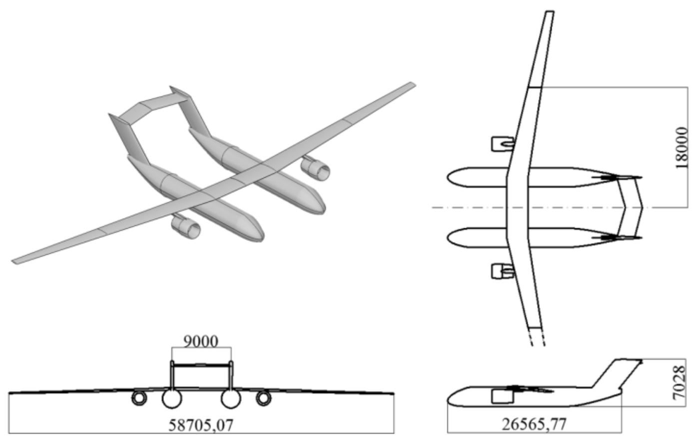

Fig. 11. Geometric dimensions of the MR-TF aircraft.

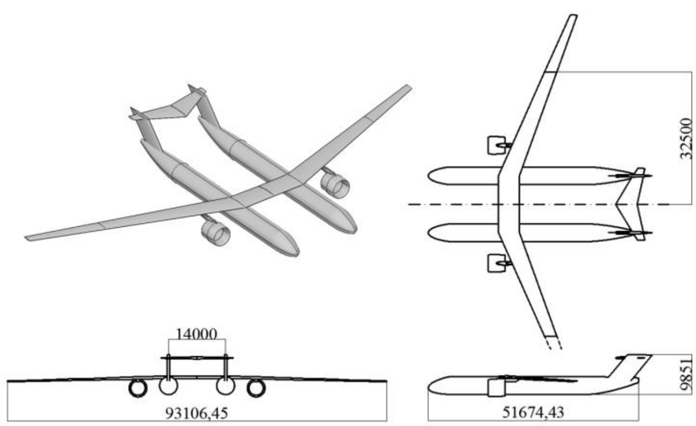

Fig. 12. Geometric dimensions of the LR-TF aircraft.

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As illustrated in Fig. 8, the initial sized aircraft were input into the modified SUAVE for performance analysis and assessment with the presented wing mass estimation method. After iterative calculations until convergence, the weight breakdown and mission segments were obtained, as shown in Table 4 and Fig. 13.

Table 4. Weight breakdown summary of the MR-TF and LR-TF aircraft

Parameter	MR-TF	LR-TF
Max. takeoff weight, kg	59,218	243,926
Max. zero fuel weight, kg	45,737	157,328
Fuel weight, kg	13,480	86,597
Empty weight, kg	31,517	124,148
Empty weight breakdown		
Wing, kg	5,755	39,323
Fuselages, kg	5,241	20,596
Propulsion, kg	3,798	17,292
Nacelles, kg	495	2,392
Landing gear, kg	2,021	6,554
Horizontal tail, kg	775	1,565
Vertical tail, kg	850	2,559
Paint, kg	418	1,211
Systems, kg	12,164	32,657

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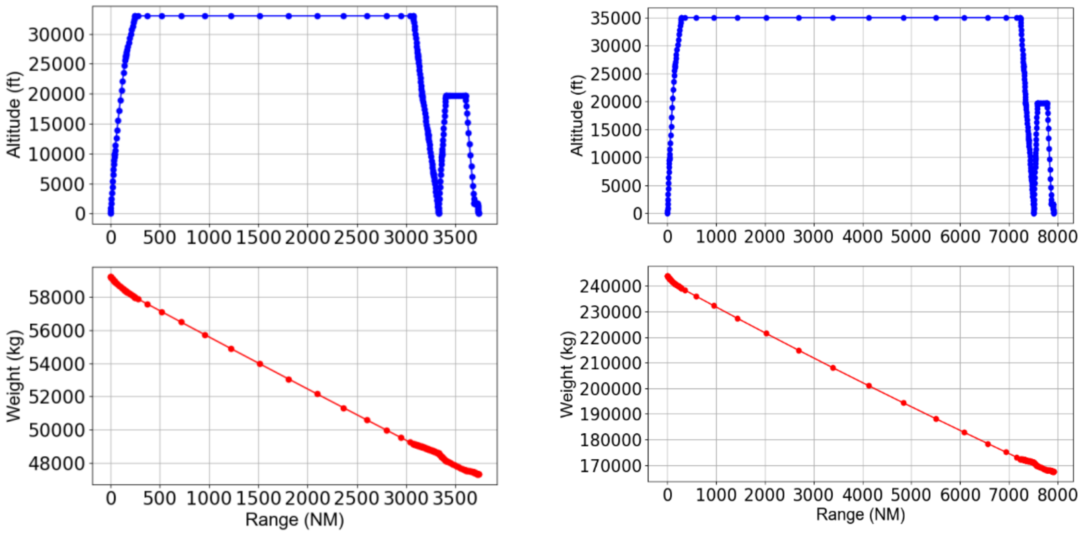

a) MR-TF b) LR-TF

Fig. 13. Mission profiles of the MR-TF and LR-TF aircraft.

4. Conclusion

This paper addressed the initial weight estimation of twin-fuselage aircraft at the conceptual design stage. A semi-analytical wing mass estimation method for the TF configuration was improved in this work, including improved fidelity of the aerodynamic analysis and extensions for advanced composite materials. Since the semi-analytical method requires a large number of input parameters, which are difficult to obtain at the initial sizing stage, DoE and regression methods were used to establish a semi-empirical wing mass estimation method for the TF configuration with different parameters' values for different missions and materials, separately. Eventually, the established semi-empirical weight estimation method was integrated into a TF aircraft conceptual design and analysis procedure and two TF aircraft, including a mid-range TF aircraft and a long-range TF aircraft, were designed and sized to demonstrate the TF aircraft design process and its weight estimation method.

The case study showed that the established initial weight estimation method for the TF aircraft configuration can efficiently analyze the weight breakdown of the TF aircraft within the presented TF aircraft sizing and analysis framework. Since the TF configuration is suitable for utilizing UHARW design,

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avenues for future work include improving the semi-analytical wing mass method to account for the aeroelasticity and flutter characteristics to develop a semi-empirical wing mass estimation method that is also sensitive to the UHARW characteristics.

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